[redacted]

RESEARCH MEMORANDUM

for the

Air Materiel Command, U. S. Air Force

PERFORMANCE OF AXIAL-FLOW SUPERSONIC COMPRESSOR OF XJ55-FF-1 TURBOJET ENGINE

III - OVER-ALL PERFORMANCE OF COMPRESSOR

By Melvin J. Hartmann and Edward R. Tysl

Lewis Flight Propulsion Laboratory
Cleveland, Ohio

CLASSIFIED DOCUMENT

This document contains classified information affecting the National Defense of the United States within the meaning of the Espionage Act, USC 50:31 and 32. Its transmission or the revelation of its contents in any manner to an unauthorized person is prohibited by law. Information so classified may be imparted only to persons in the military and naval services of the United States, appropriate civilian officers and employees of the Federal Government who have a legitimate interest therein, and to United States citizens of known loyalty and discretion who of necessity must be informed thereof.

REVIEWED BUT NOT EDITED

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON

[redacted]

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

for the

Air Materiel Command, U. S. Air Force

PERFORMANCE OF AXIAL-FLOW SUPERSONIC COMPRESSOR OF XJ55-FF-1 TURBOJET ENGINE

III - OVER-ALL PERFORMANCE OF COMPRESSOR

By Melvin J. Hartmann and Edward R. Tysl

SUMMARY

An investigation was conducted to determine the performance characteristics of the rotor and inlet guide vanes used in the axial-flow supersonic compressor of the XJ55-FF-1 turbojet engine. Outlet stators used in the engine were omitted to facilitate study of the supersonic rotor.

The extent of the deviation from design performance indicates that the design-shock configuration was not obtained. A maximum pressure ratio of 2.26 was obtained at an equivalent tip speed of 1614 feet per second and an adiabatic efficiency of 0.61. The maximum efficiency obtained was 0.79 at an equivalent tip speed of 801 feet per second and a pressure ratio of 1.29. The performance obtained was considerably below design performance. The effective aerodynamic forces encountered appeared to be large enough to cause considerable damage to the thin aluminum leading edges of the rotor blades.

INTRODUCTION

At the request of the Air Materiel Command, U. S. Air Force, the NACA Lewis laboratory is conducting an investigation of the performance characteristics of the axial-flow supersonic compressor of the XJ55-FF-1 turbojet engine. The inherent high pressure ratio per stage and high mass-flow characteristics of the supersonic-type compressor have been demonstrated in several supersonic-compressor investigations reported (references 1 to 3). These characteristics of the supersonic-type compressor, which result in a light weight, compact compressor component, are extremely desirable, especially for this engine.

The blade sections used on the compressor rotor of the XJ55-FF-1 supersonic compressor are based on design methods developed at the NACA Langley laboratory. Aerodynamic and thermodynamic design data for this compressor are given in reference 4. The engine-compressor configuration consists of a row of inlet guide vanes to provide the desired relative Mach number distribution, a rotor designed to operate at supersonic relative velocities, and two rows of stator blades downstream of the rotor. Preliminary performance of this compressor without the two rows of stator blades is presented in reference 5. The inlet guide vanes for the XJ55-FF-1 compressor have been investigated as a separate component; except for a small portion of the passage at the root and the tip, the measured turning angle approximated design (reference 6).

The investigation reported herein was conducted to determine the over-all performance characteristics of the supersonic compressor of the XJ55-FF-1 turbojet engine with outlet stators omitted. The over-all results are based on data obtained in the inlet tank and at a rake station approximately 6 inches downstream of the rotor.

APPARATUS

Variable-component compressor test stand. - Air was supplied from the laboratory refrigerated air system through the air-measuring orifice to a depression tank 4 feet in diameter and 6 feet long. A mechanical filter consisting of filter paper with a 120-mesh screen on each side and supported by a coarse mesh screen was used in the depression tank. The air flowed from the tank through a long conical adapter section to the compressor inlet.

A diagrammatic cross section of the compressor installation is shown in figure 1. The rotor was straddle-mounted, with the shaft supported on two self-alining journal bearings. The passage contour upstream of the inlet guide vanes was dictated by the section containing the four struts making up the front bearing support. The hub section upstream of the compressor rotor was supported by the inlet guide vanes and the front bearing support.

In order to evaluate the rotor performance, outlet stator blades were not installed. The hub section downstream of the rotor was supported by six airfoil struts set at an angle corresponding to the design angle of discharge at the pitch section. Survey data indicate that the losses in these struts (located about 1 in. downstream of the rotor) were small and thus the over-all performance based on the readings obtained downstream is representative of the compressor performance. Air discharged from the compressor rotor through an annular passage into the variable-component collector and then through two radial outlet pipes to the laboratory altitude exhaust facilities.

Compressor-inlet pressure was regulated by a butterfly throttle valve located between the air-measuring orifice and the inlet tank. Outlet pressure was regulated by a butterfly valve in the outlet pipe. The compressor was driven by a 2500-horsepower direct-current motor through a speed increaser. Motor speed was controlled by an electronic regulating device.

Instrumentation. - The air weight flow through the compressor was measured by a calibrated adjustable orifice in the inlet pipe. Compressor-inlet conditions were measured in the depression tank 12 inches downstream of the mechanical filter by four thermocouples and four static-pressure taps installed as recommended in reference 7. Static and total conditions can be considered equivalent because the velocity of the air was negligible in the depression tank.

Immediately downstream of the rotor a calibrated combination survey instrument was used to measure direction of flow and static and total pressures, which will subsequently be used to evaluate passage performance of the rotor and guide vanes. The static-pressure tap of the combination survey instrument was calibrated over the range of Mach numbers encountered in operation. The aerodynamic center of the claw tubes of the combination survey instrument was determined in a high-velocity jet and installed on a line parallel with the axis of the rotor. The survey tube was mounted in a remotely operated motor-driven survey mechanism that provided separate control of rotation and translation of the tube. This survey instrument and the remotely controlled actuator are described in reference 1.

Two total-temperature rakes and two total-pressure rakes alternately spaced 90° apart circumferentially were located approximately 6 inches downstream of the rotor and about 3 inches downstream of the supporting struts. Each rake consisted of three thermocouple or total-pressure tubes located at the centers of equal annular areas. Both the total-temperature and total-pressure rakes were insensitive to yaw over a range of $\pm 30^{\circ}$; the rakes could therefore be set at approximate angles of flow and accurate readings obtained over the range of conditions encountered.

All pressures were measured with mercury manometers and all temperatures were taken with iron-constantan thermocouples calibrated for recovery coefficients over the range of Mach numbers encountered. The difference in potential between the hot-junction and the ice-bath thermocouple was measured with a calibrated potentiometer in conjunction with a spotlight galvanometer. The compressor-rotor speed was measured with an electric chronometric tachometer.

<u>Compressor.</u> - The compressor assembly investigated consists of a row of inlet guide vanes and the supersonic rotor (fig. 1). The 27 blades of the inlet guide vanes have constant hub and tip diameters of 10.25 and 13.25 inches, respectively. The rotor has 38 blades with a tip diameter of approximately 13.00 inches and a hub diameter varying from 10.20 inches at the inlet to 10.65 inches at the outlet.

The compressor-rotor blades were made of high-strength aluminum alloy machined to leading-edge wedge angles varying from 8° to 12° (fig. 2). Numerous inspections of the compressor rotor were made during operation; after about 35 hours, considerable damage to the leading edges of the compressor-rotor blades was observed. This damage was in the form of a rolling-over of the thin leading edges (fig. 3(a)). As shown in the enlarged view (fig. 3(b)), no impact marks could be found on the rolled over portions of the blades. This damage is apparently the result of aerodynamic forces. Maximum steady-state aerodynamic forces would exert forces of the order of only one-fifth of the yield strength of this material. Thus it appears that the forces necessary to cause this damage would of necessity result from unsteady conditions, which could produce rather violent forces accompanying flows perpendicular to the sharp leading edges. The rolling-over of the edges eliminates a high-frequency vibration because such a vibration would break the aluminum sections.

The rotor blades were reworked by hand-filing the damaged portions of the leading edges, thus moving back the leading edges but maintaining design area ratio and leading-edge angle. The compressor was operated for about 12 hours before similar damage was observed. The second damage was only a little more extensive than that shown in figure 3(a). The rotor blades were again refinished in the same manner; after refinishing, the compressor was operated for only about 2 hours when the more extensive damage shown in figure 4(a) was observed. As shown in the enlarged view (fig. 4(b)), the leading edges were again rolled over but in this case the damage extended along the entire length of the blade. After the two hand-filings and about 47 hours of operation, these portions of the blades had been considerably work-hardened, resulting in the more extensive damage.

Metallurgical examination showed that the material used in these blades had very strong directional characteristics. The crystalline structure of the blade examined was such that very poor fatigue life would result in direction of failure. This characteristic of the blade material was probably the cause of the increased extent of damage and the reduced operating time in the two failures, but it is not considered sufficient to explain the cause of this type of failure.

PROCEDURE

Performance data were obtained over a range of equilvalent weight flows $W\sqrt{\theta}/\delta$ (where θ is ratio of atmospheric temperature to standard sea-level temperature and δ is ratio of atmospheric pressure to standard sea-level pressure) from wide-open throttle to stall at constant equivalent tip speeds $U/\sqrt{\theta}$ ranging from 409 to 1614 feet per second. Adiabatic efficiency η_{ad} based on the arithmetically averaged total temperature and total pressure at the rake station was computed, as described in reference 1.

The over-all performance was first obtained up to an equivalent tip speed of 1498 feet per second. Before data could be taken at any higher speed the leading-edge damage to the rotor blades was observed. After refinishing the rotor-blade leading edges, several readings were taken at an equivalent tip speed of 1396 feet per second, which showed that the maximum total-pressure ratio checked the previous results within $1\frac{1}{2}$ percent. The data at 1614 feet per second were obtained at this point; the first refinishing of the rotor blades therefore had only a small effect on the over-all performance of the rotor. Performance data were not obtained after the second refinishing of the blade leading edges.

Survey data obtained downstream of the compressor rotor were obtained to determine passage flow conditions. These data were used to obtain an integrated mass flow, which checked the orifice-measured weight flow within 2 percent. Total-pressure ratios obtained from the survey instrument were approximately 1 percent higher than those obtained from the rakes downstream of the supporting struts. The losses in the supporting struts thus did not have an appreciable effect on the results obtained; over-all results presented herein are based on the downstream-rake data.

RESULTS AND DISCUSSION

The over-all compressor data obtained in this investigation are shown in figure 5. Total-pressure ratio is shown as a function of equivalent weight flow with superimposed contours of adiabatic efficiency for eight equivalent tip speeds in the range from 409 to 1614 feet per second. The range of pressure ratios at equivalent tip speeds of 1498 and 1614 feet per second are incomplete because of the damage to the rotor-blade leading edges; the dashed performance lines shown in the lower range of pressure ratios can be justified by extrapolation of the trends in the performance curves up to 1396 feet per second.

Maximum performance at 1614 feet per second showed a pressure ratio of 2.26 and an efficiency of 0.61. A maximum efficiency of 0.79 was obtained at an equivalent tip speed of 801 feet per second and a pressure ratio of 1.29. In the lower speed range (409, 587, and 801 ft/sec) steady operation could be maintained over a wide range of equivalent weight flows. As the speed was increased, the equivalent-weight-flow range was decreased until at 1396 feet per second the pressure ratio was varied over the range from open throttle to stall with very little change in equivalent weight flow. Thus the performance curves at 1498 and 1614 feet per second must be essentially vertical lines on the performance map of figure 5.

As indicated in the over-all performance map (fig. 5), the adiabatic efficiency decreased from 0.79 to 0.61 as the equivalent tip speed was increased from 801 to 1614 feet per second. This decrease is somewhat in contrast to the hypothesized effect discussed in reference 3 (and essentially borne out in material published in reference 2) where it is stated that the adiabatic efficiency of a supersonic compressor could be expected to remain essentially constant with speed.

Comparing the maximum measured equivalent weight flow obtained with that predicted by theory indicates that the equivalent weight flows are about 17.5 percent less than design. It can be noted that the results presented in reference 2 showed equivalent weight flows (at the rotor speed at which supersonic operation was reached) that were only $3\frac{1}{2}$ percent less than that predicted.

The reasons for this loss in performance cannot be completely determined from an inspection of over-all performance. The low equivalent weight flow and the extreme loss in efficiency with increase in speed indicate, however, that the design-shock configuration was not obtained.

1168

It is significant that this compressor represents the first attempt to produce a supersonic compressor on a commercial basis and that the design pressure ratio of 2.845 is somewhat higher than has been previously attained. The compressor investigations reported in references 1 to 3 have shown that the shock configuration is extremely sensitive to small changes in area ratio, rotor-blade and passage contours, and boundary-layer build-up. The adverse effects encountered in this investigation therefore may probably be reduced by minor alterations to the compressor. Some indication of the required changes may be obtained from a complete analysis of performance based on the surveys made downstream of the rotor.

SUMMARY OF RESULTS

The following over-all results were obtained in the investigation of the axial-flow supersonic compressor of the XJ55-FF-1 turbojet engine:

1. The magnitude of the deviation between observed performance and design performance indicates that the design-shock configuration was not obtained.

2. The maximum pressure ratio of 2.26 was obtained at an equivalent tip speed of 1614 feet per second and an adiabatic efficiency of 0.61. A maximum efficiency of 0.79 was obtained at an equivalent tip speed of 801 feet per second and a pressure ratio of 1.29.

3. The resulting aerodynamic forces, which appeared to be partly due to unsteady operation, were large enough to cause considerable damage to the thin leading edges of the aluminum rotor blades.

Lewis Flight Propulsion Laboratory,
National Advisory Committee for Aeronautics,
Cleveland, Ohio, May 27, 1949.

Melvin J. Hartmann

Melvin J. Hartmann,
Aeronautical Research Scientist.

Edward R. Tysl

Edward R. Tysl,
Aeronautical Research Scientist.

Approved:
Robert O. Bullock,
Aeronautical Research Scientist.

Oscar W. Schey,
Aeronautical Research Scientist.

jgm

REFERENCES

1. Ritter, William K., and Johnsen, Irving A.: Performance of 24-Inch Supersonic Axial-Flow Compressor in Air. I - Performance of Compressor Rotor at Design Tip Speed of 1600 Feet per Second. NACA RM E7L10, 1948.

2. Johnsen, Irving A., Wright, Linwood C., and Hartmann, Melvin J.: Performance of 24-Inch Supersonic Axial-Flow Compressor in Air. II - Performance of Compressor Rotor at Equivalent Tip Speeds from 800 to 1765 Feet per Second. NACA RM E8G01, 1949.

3. Erwin, John R., Wright, Linwood C., and Kantrowitz, Arthur: Investigation of an Experimental Supersonic Axial-Flow Compressor. NACA RM L6J01b, 1946.

1168

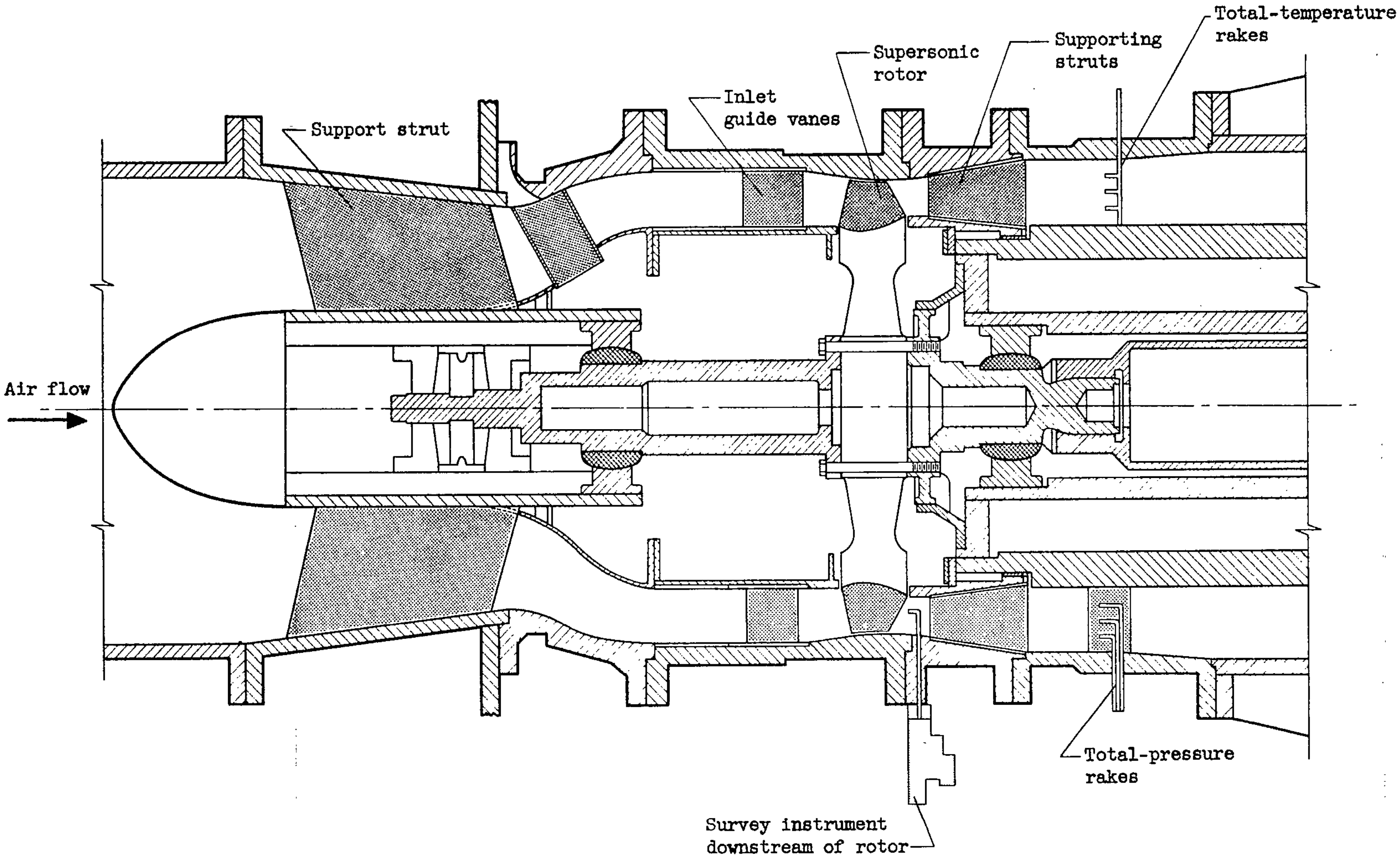
4. Blanton, J., and Dunlavy, H.: Aerodynamic and Thermodynamic Design of the Compressor for the XJ-55-FF-1 Gas Turbine. Rep. No. 124-R4, Frederic Flader, Inc. (North Tonawanda, N.Y.), Sept. 25, 1947.

5. Hartmann, Melvin J., and Graham, Robert C.: Performance of Axial-Flow Supersonic Compressor of XJ-55-FF-1 Turbine Engine. I - Preliminary Performance of Compressor. NACA RM SE9A31, U.S. Air Force, 1949.

6. Graham, Robert C., and Tysl, Edward R.: Performance of Axial-Flow Supersonic Compressor of XJ55-FF-1 Turbojet Engine. II - Performance of Inlet Guide Vanes as Separate Component. NACA RM SE9E03, U.S. Air Force, 1949.

7. NACA Subcommittee on Compressors: Standard Procedures for Rating and Testing Multistage Axial-Flow Compressors. NACA TN 1138, 1946.

[redacted]



[redacted]

Figure 1. - Diagrammatic cross-sectional view of supersonic-compressor installation.

1168

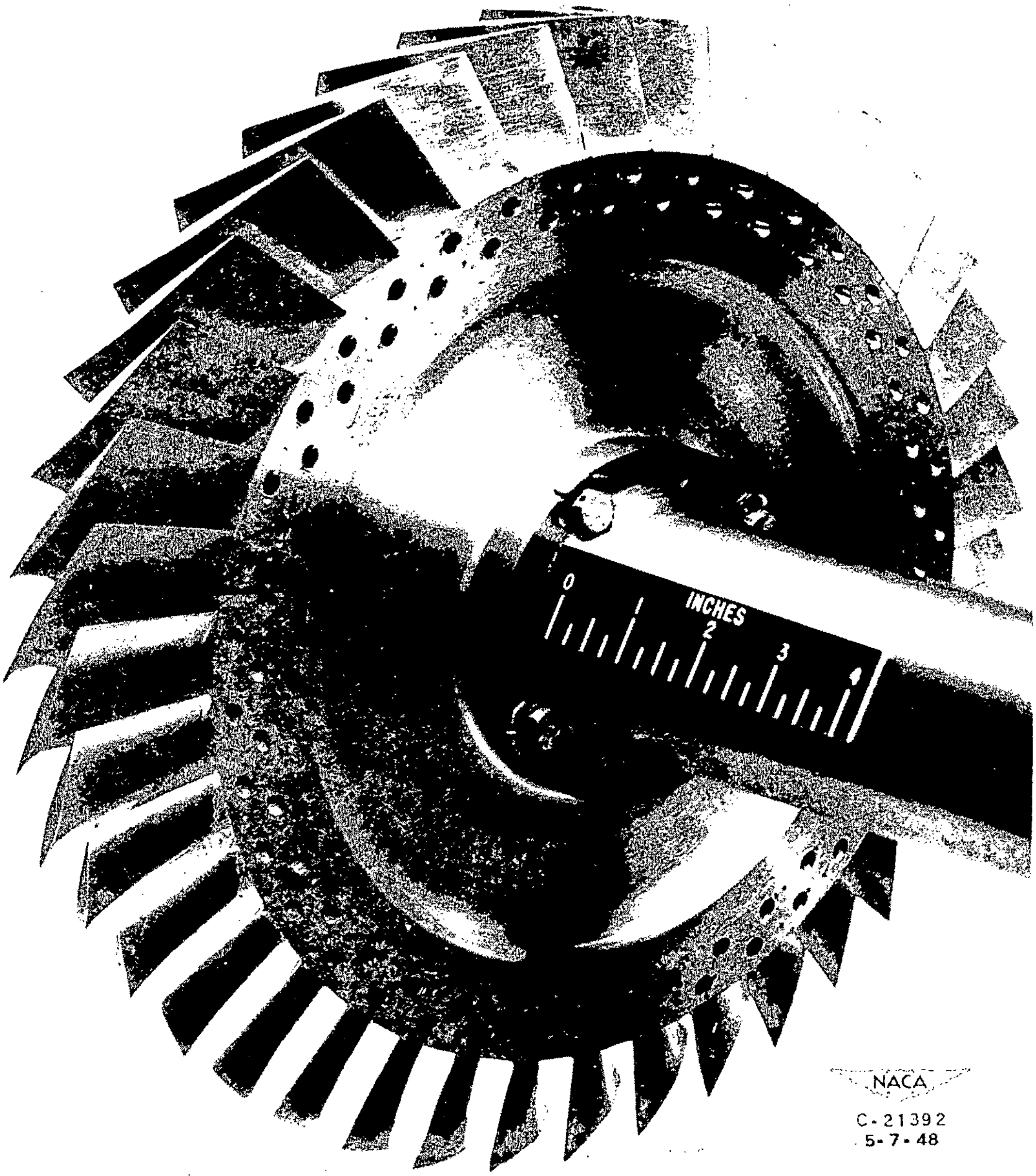
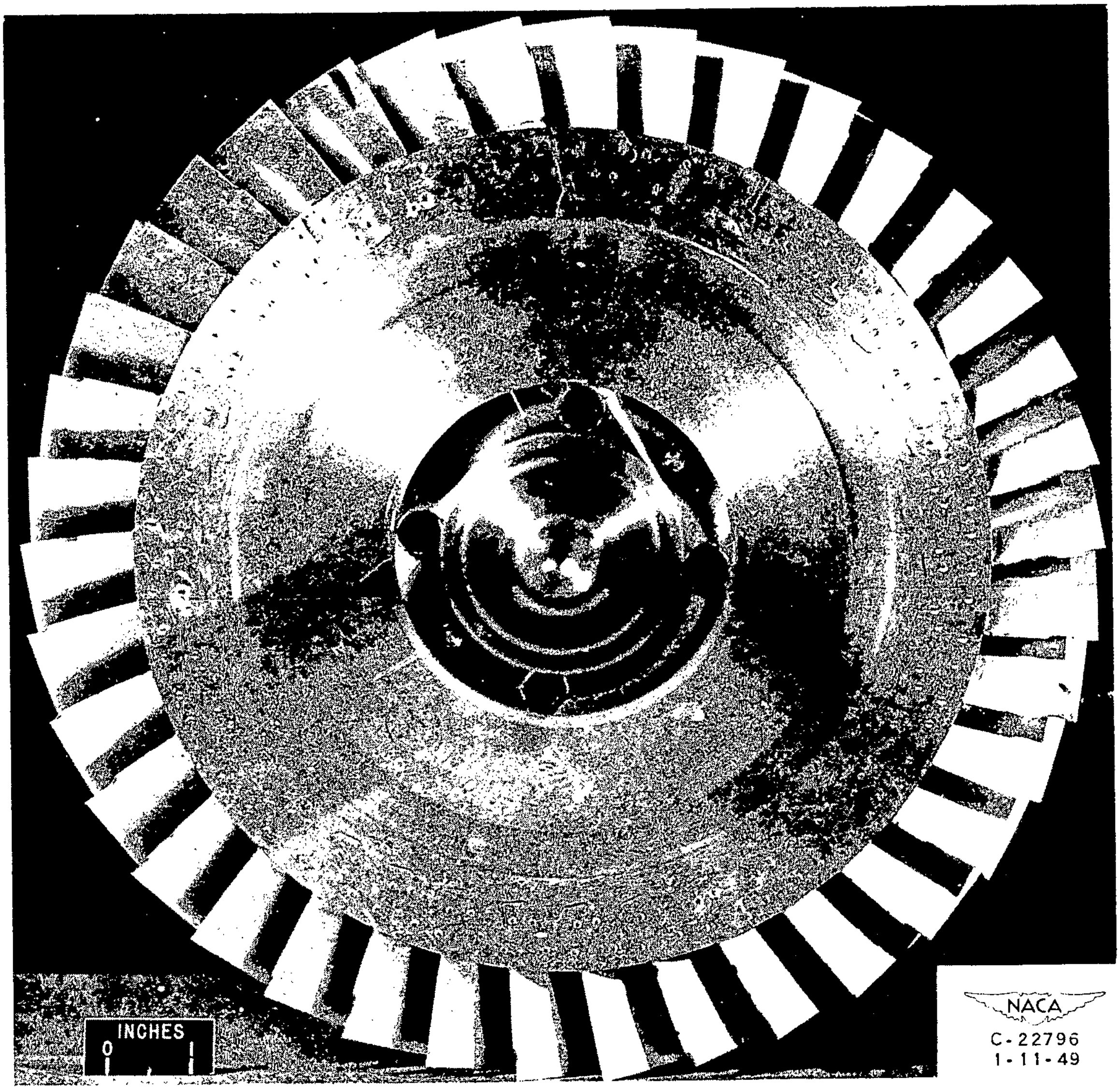


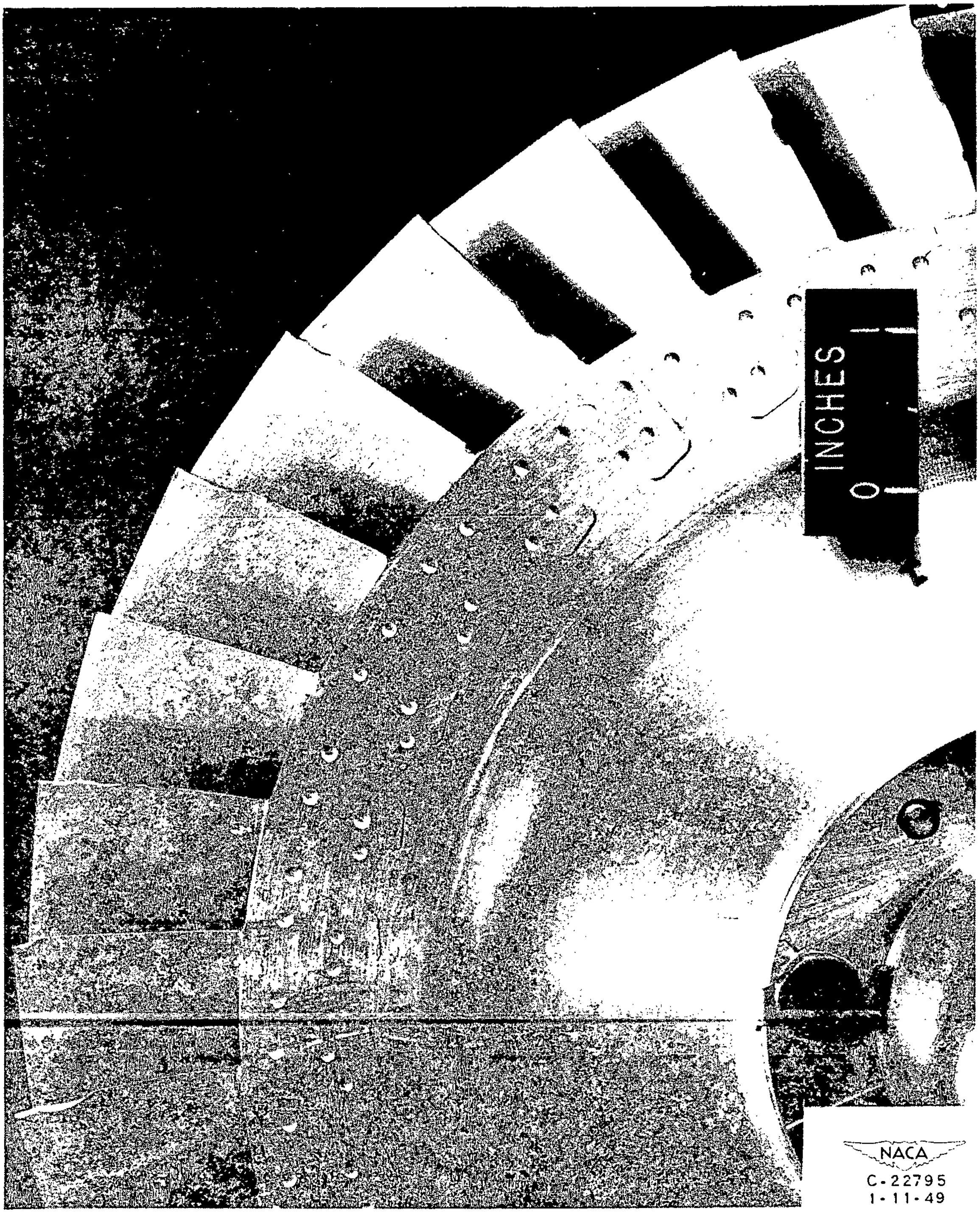
Figure 2. - Supersonic compressor rotor.

1168



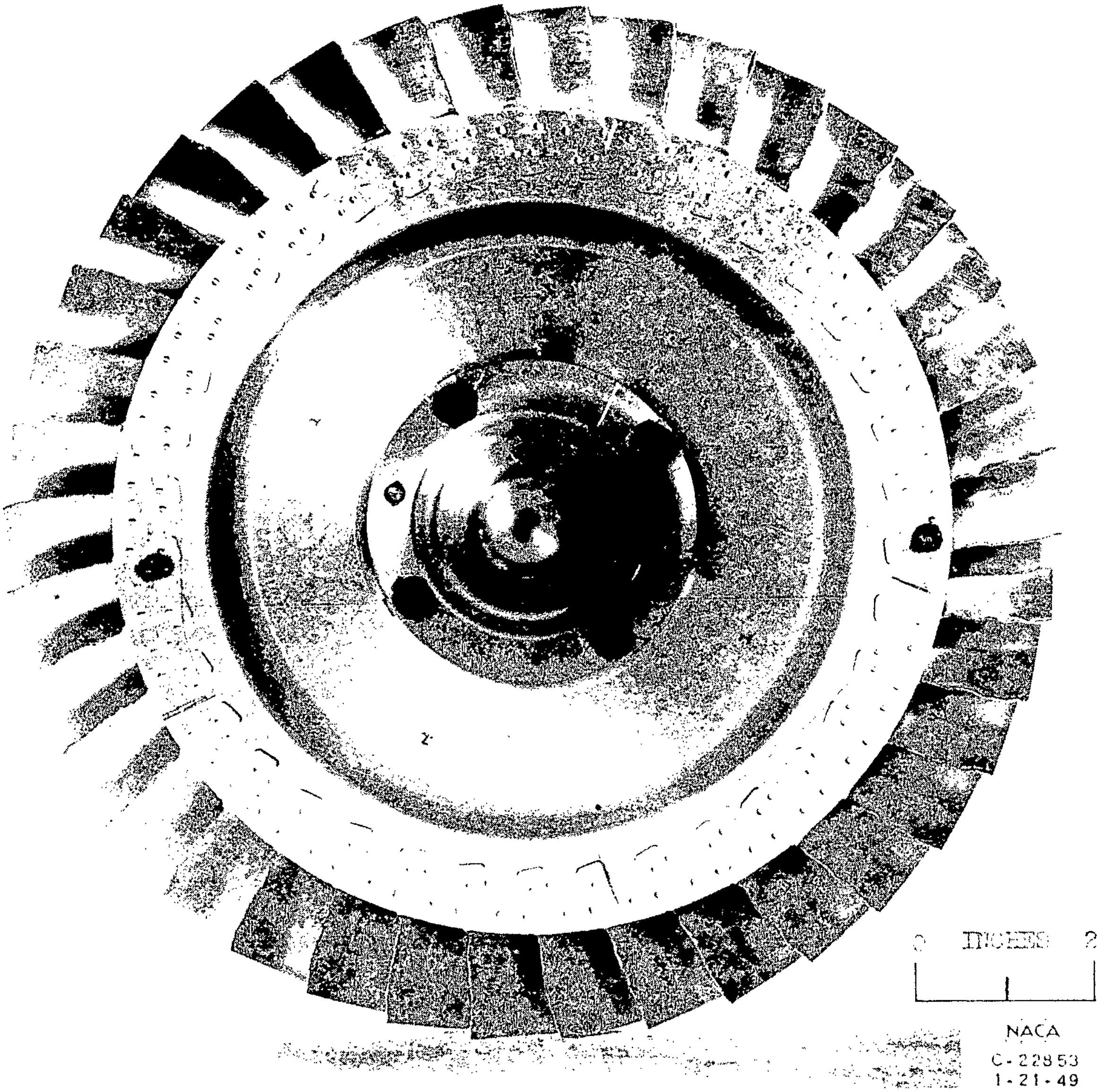
(a) Entire rotor.
Figure 3. - Initial damage to compressor rotor.

1168



(b) Enlarged view of section of rotor.
Figure 3. - Concluded. Initial damage to compressor rotor.

1168



(a) Entire rotor.
Figure 4. - Final condition of compressor rotor.

1168

(b) Enlarged view of section of rotor.
Figure 4. - Concluded. Final condition of compressor rotor.

1168

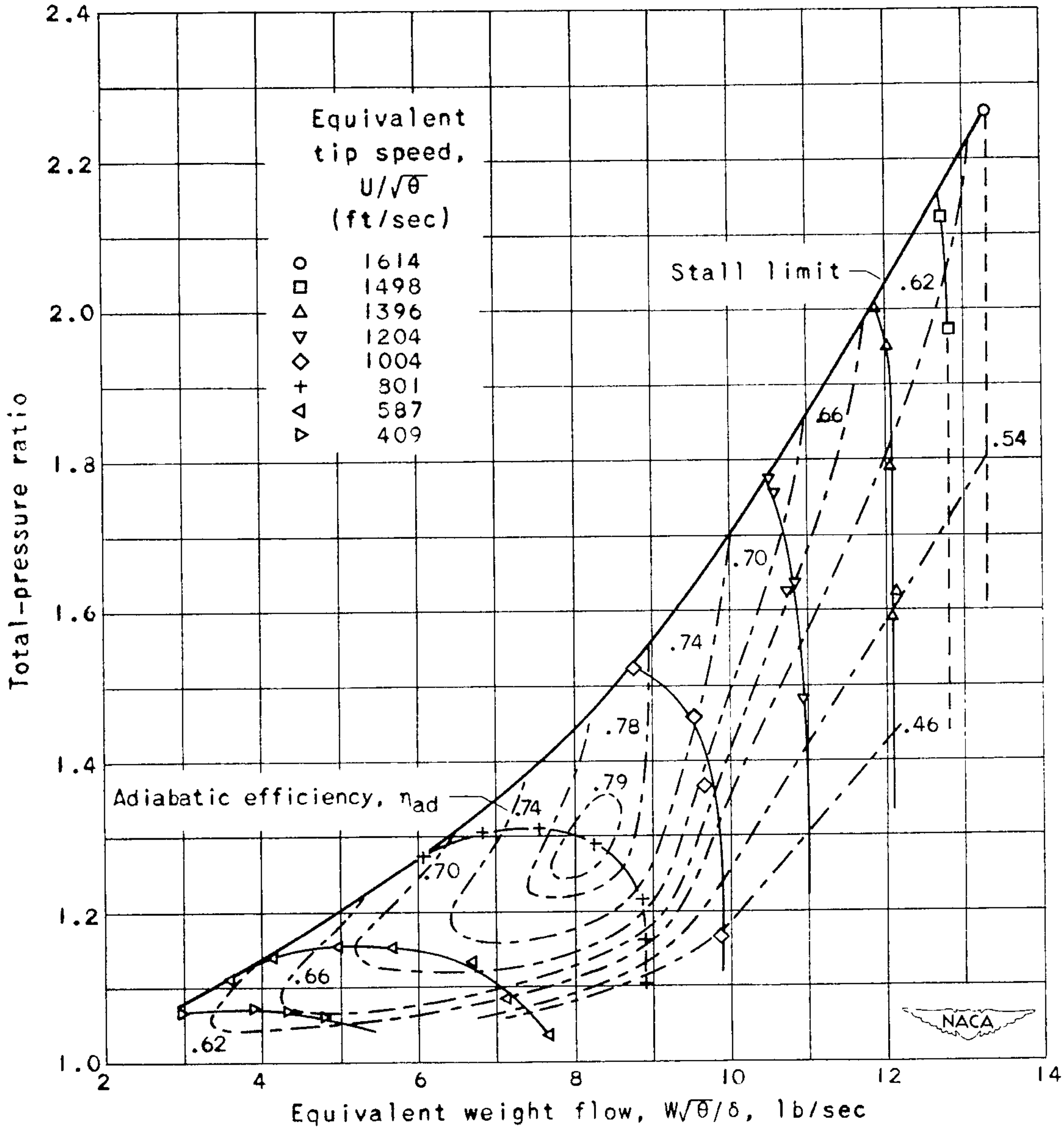


Figure 5. - Performance characteristics of axial-flow supersonic compressor for XJ55-FF-1 turbojet engine. (Fig. 6 of reference 5.)